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RESEARCH MEMORANDUM

DAMPING-IN-ROLL CHARACTERISTICS OF A 42.7° SWEPTBACK WING AS DETERMINED FROM A WIND-TUNNEL INVESTIGATION OF A TWISTED SEMISPAN WING

By

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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SUMMARY

An investigation was made to determine the damping-in-roll characteristics of a 42.7° sweptback wing by utilizing a linearly twisted wing to represent a rolling wing. The wind-tunnel tests were made using a wing of 42.7° leading-edge sweep, aspect ratio 4, taper ratio 0.5, and a 10-percent-thick circular-arc airfoil section normal to the 50-percent chord line of the unswept panel. The tests were made in the transonic speed range from a Mach number of 0.6 to 1.15 in the Langley high-speed 7- by 10-foot tunnel by utilizing the transonic bump and at a Mach number of 1.9 in the Langley 9- by 12-inch supersonic blowdown tunnel. The effect of thickening the trailing edge of the aileron on the damping-in-roll characteristics was also determined in the transonic speed range.

The damping-in-roll coefficient as determined in the present investigation agreed very well with results of tunnel tests of a free-roll rocket vehicle for Mach numbers up to 0.9. Good agreement was obtained between experimental values of the damping coefficient and theoretical values at subsonic Mach numbers. At supersonic Mach numbers the linearized theory gave values greater than the experimental.

The thickened trailing-edge contour in the normal location of the aileron gave larger values of the damping coefficient throughout the transonic speed range than the normal circular-arc contour.

INTRODUCTION

The advent of the supersonic airplane has brought to attention the scarcity of data on stability derivatives at transonic and supersonic speeds. One of these derivatives, the damping in roll C_{l_p}, is an important factor in predicting the lateral stability characteristics as well as the maneuverability of an airplane. There have been some theoretical approaches made to the problem such as those of references 1 to 3. These theories, however, do not apply in the immediate vicinity of $M = 1$ and thus experimental investigations are needed to provide design data in this speed region. The present investigation is an experimental approach to the problem and also a check on the method.

The method involved assumes that the measured rolling-moment coefficient of a wing with a linear-twist variation along the span is equal to the damping coefficient provided by a wing in steady roll.

In an investigation of this type two wings are necessary, one twisted and one untwisted. The untwisted wing provides a basis for determining the rolling moment resulting from twist.

This paper presents the results of the damping-in-roll investigation for two wing configurations, one with the normal wing contour and one with a thickened trailing edge in the normal location of the aileron. A comparison of the experimental results with theory is included in the paper.

Previous investigations of the aileron effectiveness of this wing are reported in references 4 and 5.

COEFFICIENTS AND SYMBOLS

C_l rolling-moment coefficient $\left(\dfrac{L'}{q\frac{S}{2}b}\right)$

C_{l_p} damping-in-roll coefficient $\left(\Delta C_l \times \dfrac{2V}{pb} \times K\right)$

where

q effective dynamic pressure over span of model, pounds per square foot $\left(\frac{1}{2}\rho V^2\right)$

L' rolling moment about wind axes, foot-pounds

S twice wing area of semispan model, 0.250 square foot

b twice span of semispan model, 1.000 feet

ρ mass density of air, slugs per cubic foot

V airspeed, feet per second

ΔC_l increment in rolling-moment coefficient caused by wing twist at a given angle of attack

K reflection-plane correction factor, applied to the transonic rolling-moment coefficients only

$\frac{pb}{2V}$ wing-tip helix angle, radians

p rate of roll corresponding to a given airspeed, radians per second

M effective Mach number over span of model

M_a average chordwise local Mach number

M_l local Mach number

c local wing chord

t thickness

y spanwise distance from plane of symmetry, feet

α angle of attack, degrees

δ control-surface deflection, degrees

APPARATUS AND TESTS

Apparatus

In the investigation two wings were used which were identical in geometry except for the twist of the airfoil sections along the span. The wings had a leading-edge sweepback of 42.7°, a taper ratio of 0.5, an aspect ratio of 4.0, and a 10-percent-thick circular-arc airfoil section normal to the 50-percent chord line of the unswept panel. The wings were made of steel and the half fuselage of brass. The details of the model are given in figure 1.

One wing was twisted to provide a nearly linear variation of twist along the span of the model. The results of the twisting are shown in figure 2. The 3.44° of twist at the tip corresponds to a wing-tip helix angle of 0.060 radian on a rolling wing. (The direction of twist on the model corresponds to the upgoing wing panel of a rolling airplane.) The other wing was left untwisted to provide a basis for determining the amount of rolling moment due to twist.

During the course of the investigation a modification was made to both wings which consisted of building up the trailing edge (see fig. 1) with solder in the space normally occupied by an aileron $\left(\text{from 80 percent chord to trailing edge and from } 0.50\frac{b}{2} \text{ to tip}\right)$. The resulting trailing-edge thickness was one-half the thickness of the airfoil at 80 percent chord.

The tests of the wing in the transonic speed range were conducted with the fuselage attached, whereas those at a Mach number of 1.90 were made without a fuselage.

Tests

The models were tested in the Langley high-speed 7- by 10-foot tunnel by utilizing the flow field over the transonic bump (see fig. 3) to obtain Mach numbers from 0.6 to 1.15. A five-component balance of the strain-gage type is installed beneath the surface of the bump and measures forces and moments with respect to the wind axes.

The data at a Mach number of 1.90 were obtained from the Langley 9- by 12-inch supersonic blowdown tunnel on the same wings as were used in the transonic investigation. This tunnel utilizes the exhaust air of the Langley 19-foot pressure tunnel. The model was mounted on a four-component strain-gage balance which was attached to the tunnel floor. The balance rotates through the angle-of-attack range with the model and measures normal force, chord force, pitching moment, and rolling moment due to normal force. (See reference 6 for additional details.)

Typical contours of local Mach number in the vicinity of the model location on the bump are shown in figure 4. It is seen that there was a Mach number variation of about 0.07 over the model semispan at low Mach numbers and about 0.10 at the highest Mach numbers. The chordwise Mach number variation was generally less than 0.02. No attempt has been made to evaluate the effects of this chordwise and spanwise Mach number variation. The long dashed lines shown at the root of the wing (fig. 4) indicate a local Mach number 5 percent below the maximum value and represent a point at which the boundary layer was assumed to begin.

The effective test Mach number was obtained from contour charts similar to those presented in figure 4 using the relationship

$$M = \frac{2}{S} \int_{0}^{b/2} cM_a \, dy$$

The variation of mean Reynolds number with test Mach number is shown in figure 5. The boundaries on the figure are an indication of the probable range in Reynolds number caused by variations in test conditions in the course of the investigation.

The free-stream Mach number of the supersonic tunnel has been calibrated at 1.90 ± 0.02. This Mach number was used in determining the dynamic pressure. During the tests the dynamic pressure, as well as the Reynolds number, decreased about 5 percent because of the decreased pressure of the inlet air. The average Reynolds number during a test was about 2.2×10^6. The various factors which might affect the test results of this tunnel are discussed in reference 6. One of the factors which might conceivably affect the results is the tunnel boundary layer which is about 0.4 inch thick; however, the effect of this factor should be rather small because most of the rolling moment comes from the loading at the tip.

The assumption that a rolling wing is correctly represented by a wing with a linear-twist distribution along the span is not quite accurate. The pressure distribution over the twisted wing is the result of the moving air stream, whereas the pressure distribution over the rolling wing is the result of the forward velocity and the rolling velocity of the airplane. However, the error involved in computing C_{l_p} by using the free air-stream velocity is less than 1 percent.

A reflection-plane correction which accounts for the carry-over of load to the other wing panel has been applied to the twisted-wing data. This correction factor (0.857) was determined by using span loading computed by Weissenger's method and accounts for sweep but not for Mach number. The error caused by neglecting the effects of Mach number on the semispan correction is alleviated somewhat if the data are used in conjunction with aileron data obtained by the same method and corrected in the same manner. If the values of C_{l_p} are to be used in lateral-stability calculations, this compensating effect is not obtained and thus the values represent the trend with Mach number but may be in error as to the absolute magnitude.

RESULTS AND DISCUSSION

Experimental Results

The damping in roll was determined from plots of the rolling-moment coefficients against angle of attack similar to those curves which appear in figure 6. Figure 7 shows a typical variation of the rolling-moment coefficients through the transonic speed range for the wings with the thickened trailing-edge contour.

The results of the tests are presented in figure 8 from which it is evident that the airfoil contour has a considerable effect on the damping coefficient in roll both in magnitude and in variation with Mach number. The large difference in C_{l_p} noted throughout the Mach number range was attributed partly to the large difference of lift-curve slope that exists between the two configurations, as is shown in figure 9. (The data of fig. 9 are from another investigation of this wing, the results of which have not been published.) The lift-curve slopes from both models show similar variations with Mach number, but variations of C_{l_p} with M are not similar. The normal-contour airfoil shows little variation of C_{l_p} with M below a value of 0.85; whereas with the thickened trailing-edge model, C_{l_p} shows a considerable increase. Thickening the trailing edge shifts the Mach number at which the sudden drop-off in C_{l_p} occurs from 0.85 to 0.94. It is thought that separation occurs on the normal-contour wing near the aileron hinge line, whereas the thickened trailing edge tends to fill up the region of separation or reduced pressure gradient over the trailing edge and thus prevent or delay separation. These data are in agreement with a previous investigation of the aileron characteristics (reference 4) which showed the normal aileron to reverse effectiveness at approximately the same Mach number as the minimum value of C_{l_p} occurs.

The value of the damping-in-roll coefficient obtained at a Mach number of 1.9 is -0.233 (fig. 8), which appears to be relatively low with respect to the values obtained in the transonic speed range. It is also noted, however, that the lift-curve slope is correspondingly lower than the values in the transonic-speed range.

Figure 10 shows that the experimental results for the thickened trailing edge of the present investigation agrees very well with results obtained on a free-to-roll three-wing missile (unpublished data). The variation of C_{l_p} with Mach number for the twisted-wing model is very similar to that obtained on the larger three-wing missile for the comparable Mach numbers, and the actual values of C_{l_p} are only slightly larger for the twisted wing than for the missile. (The tests of the missile were performed in the Langley high-speed 7- by 10-foot tunnel on a regular RM-5 test vehicle where both the static rolling moment and the rate of roll are measured.)

An estimation of the variation of wing-tip helix angle $pb/2V$ with Mach number has been made (fig. 11) for the model with the thickened trailing edge by using the aileron effectiveness given in reference 4. With the exception of a Mach number of approximately 0.95, the twisted-wing results are in good agreement with the $pb/2V$ obtained from free flight of an RM-5 test vehicle (reference 7).

This relatively large discrepancy is believed, in part, to be caused by the Mach number variation on the bump that would tend to smooth out curves exhibiting abrupt changes with Mach number because each section would tend to reach its critical Mach number at a different speed. In addition, these values of $pb/2V$ were estimated from zero-angle-of-attack data, whereas the missile wing was operating at an angle of attack as a result of the rolling. There were not sufficient data to account for this effect, but the data that are available (reference 4) indicate considerable decrease in aileron effectiveness with angle-of-attack increases in that speed range. The large difference in Reynolds number between the RM-5 missile and the bump results may also have an effect in this speed range. (The Reynolds number of the missile varied from 1.6×10^6 at $M = 0.6$ to 4.1×10^6 at $M = 1.2$.)

The value of $pb/2V$ (0.002 per degree) obtained with the RM-5 missile at a Mach number of 1.9 (reference 7) is in excellent agreement with the value estimated (0.002 per degree) by using the aileron effectiveness of reference 6 and the damping coefficient of figure 8.

In normal flight, the airplane will be operating at some positive angle of attack and a knowledge of how the damping varies with airplane attitude is desirable. Figure 12 presents the variation of damping coefficient with angle of attack for a Mach number of 0.8 which was considered a typical case for Mach number range between 0.6 and 0.95. It is noted that the damping is constant between $\alpha = \pm 2^\circ$ and that a maximum value is attained at about $\alpha = 6^\circ$. The maximum values appear to be about 35 to 45 percent greater than that between $\alpha = \pm 2^\circ$. At Mach numbers between 1.0 and 1.15 the damping up to an angle of attack of 8° is approximately constant, but above this angle of attack it drops off.

Theoretical Results

A theoretical investigation was made to determine the damping in roll throughout a range of Mach numbers from 0.6 to 1.9. Figure 13 presents the results of this investigation as well as a comparison with the experimental results. Several considerations are necessary to compute C_{l_p} in the supersonic range because at certain Mach numbers various parts of the wing will be affected by subsonic or supersonic flow. Some of this theory is yet to be made available. The subsonic theory applies only up to the critical Mach number which, for this case, was taken to be $M = 0.85$, the point at which C_{l_p} tends to decrease.

The values of C_{l_p} in the subsonic range were computed by the method given in reference 1 for the normal-contour wing. This method is based on lifting-line theory in which the lifting-surface-theory correction factors to account for the effect of sweep have been applied to the lifting-line theory. In this method the section lift-curve slope was used in the calculation of C_{l_p} so that a first-order approximation is made for the thickness of the airfoil. The comparison made in figure 13 of experimental results and theory in the subsonic range shows good agreement.

The method of reference 2, which is based on linearized supersonic flow theory, was used to compute C_{l_p} for the Mach number from 1.15 to 1.36. These Mach numbers are the limits for which the wing is wholly contained within the Mach cones springing from the wing apex and from the trailing edge of the root section.

Figure 13 shows a considerable difference in the damping coefficient between experimental results and theory at a Mach number of 1.15. Part of this difference is probably the result of the airfoil section considered. The predicted values are based on thin airfoil theory, whereas the experimental results are based on a section 10 percent thick. Supporting evidence that thickness affects the damping is the change in lift-curve slopes shown in figure 9 as a result of modifying the airfoil contour. Other recent investigations have shown that the lift-curve slope cannot be predicted closely by linearized theory; hence the damping-in-roll coefficient cannot be predicted with any great degree of accuracy.

The damping coefficient determined experimentally at a Mach number of 1.9 (fig. 13) is considerably smaller than that which the linearized theory of reference 3 provides and again part of the difference may be attributed to section thickness considered. The theory is limited to the case where the leading edges lie ahead of the Mach cones emanating from the leading edges of the tip and the center line.

CONCLUSIONS

The results of tunnel tests of a twisted semispan wing to determine the damping-in-roll characteristics of a wing with 42.7° leading-edge sweep, aspect ratio 4, and taper ratio 0.5 at transonic speeds and at a Mach number of 1.9 indicated the following conclusions:

1. At subsonic speeds the damping coefficient determined by the twisted-wing method agreed very well with that obtained from tunnel tests of a free-to-roll three-wing test vehicle.

2. The wing with the thickened trailing edge in the normal location of the aileron gave larger values of damping coefficient than the normal circular-arc-contour wing. The difference was partly attributed to the large change in lift-curve slope which accompanied the modification.

3. The damping coefficients obtained theoretically at subsonic speeds were in general agreement with experimental results. At supersonic speeds, however, the linearized theory gave values which were considerably greater than the experimental results.

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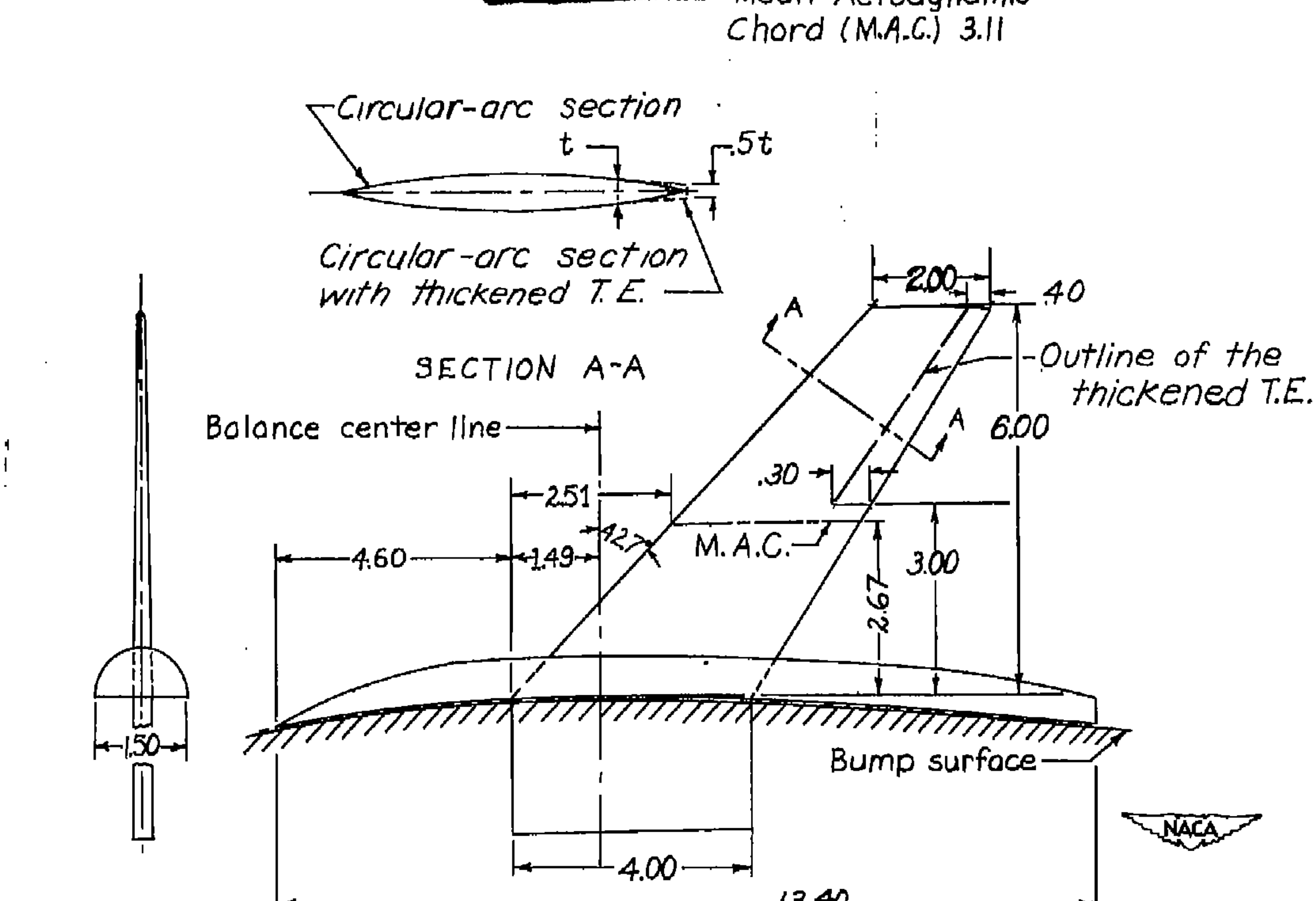

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Figure 1.- General arrangement of model. Leading-edge sweepback 42.7°, aspect ratio 4.0, taper ratio 0.5, and 10-percent-thick circular-arc airfoil section normal to 0.50 chord of unswept panel.

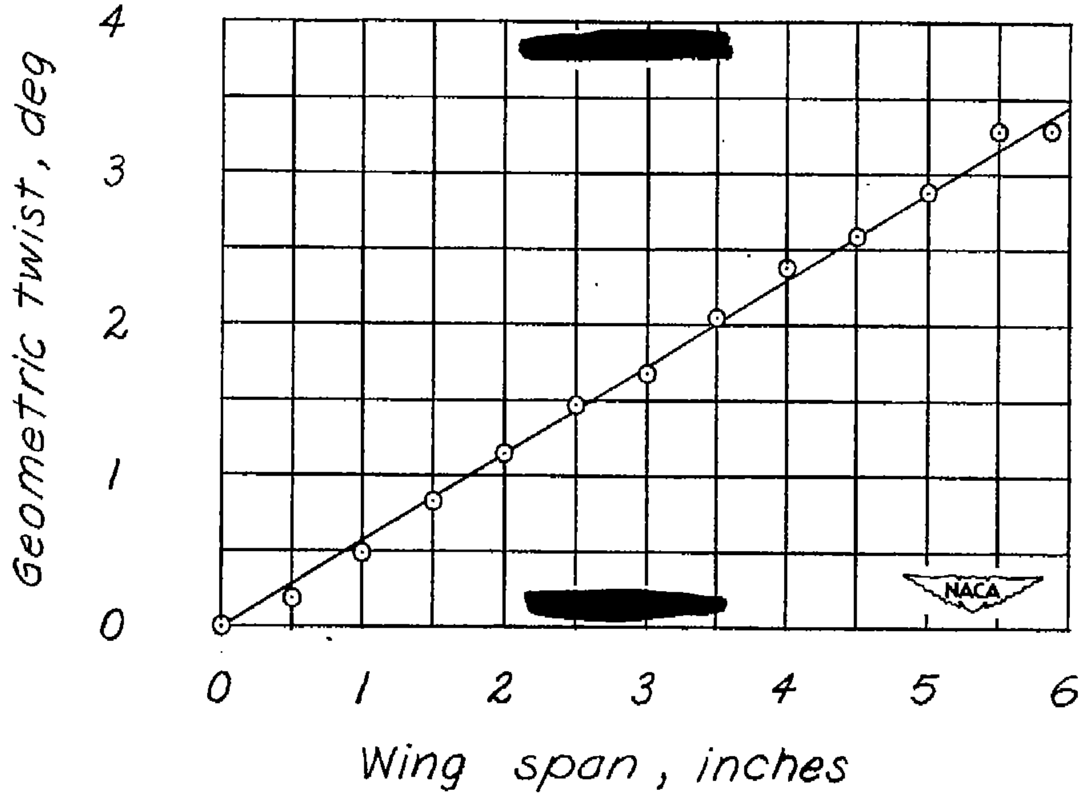

Figure 2.- Variation of geometric twist along span of model.

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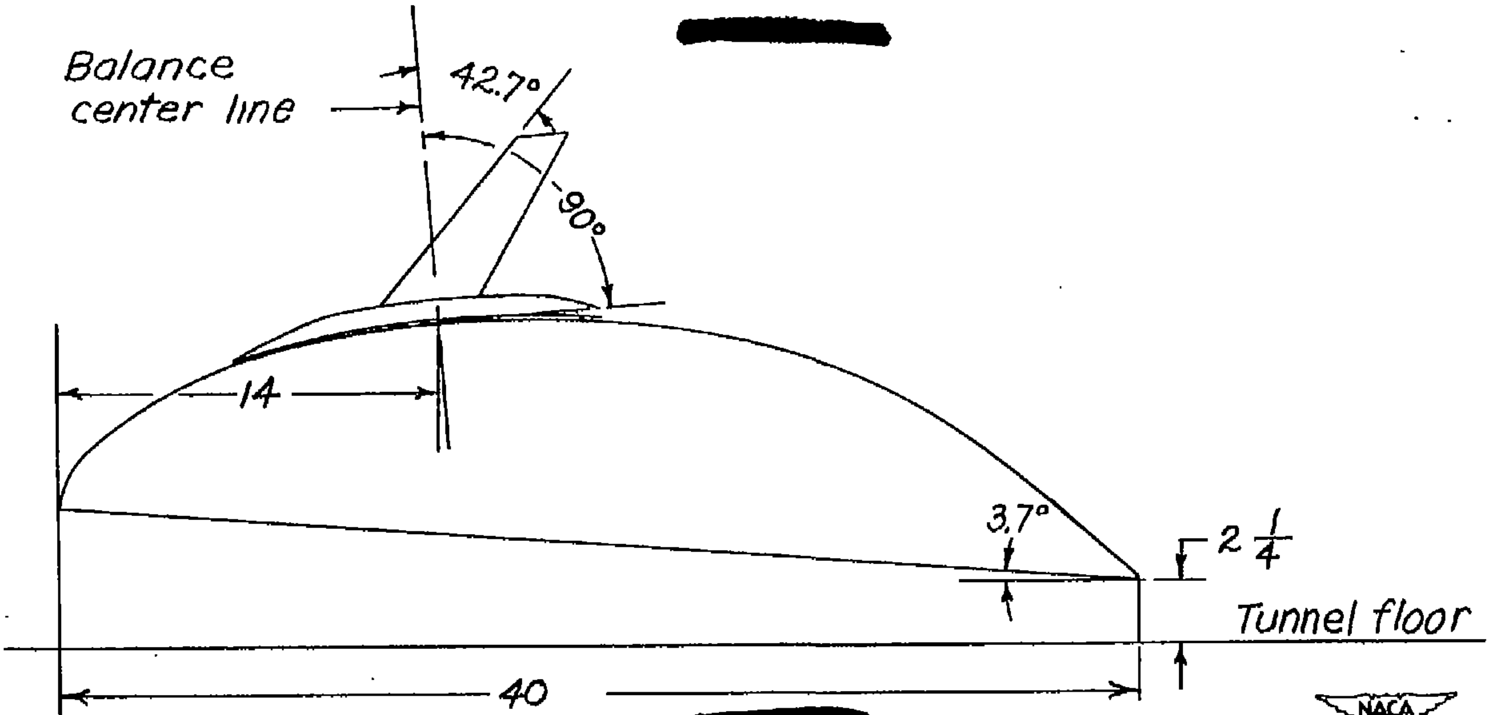

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Figure 3.– Schematic sketch of relative position of model, balance, and transonic bump as mounted in the Langley high-speed 7- by 10-foot tunnel. Dimensions in inches.

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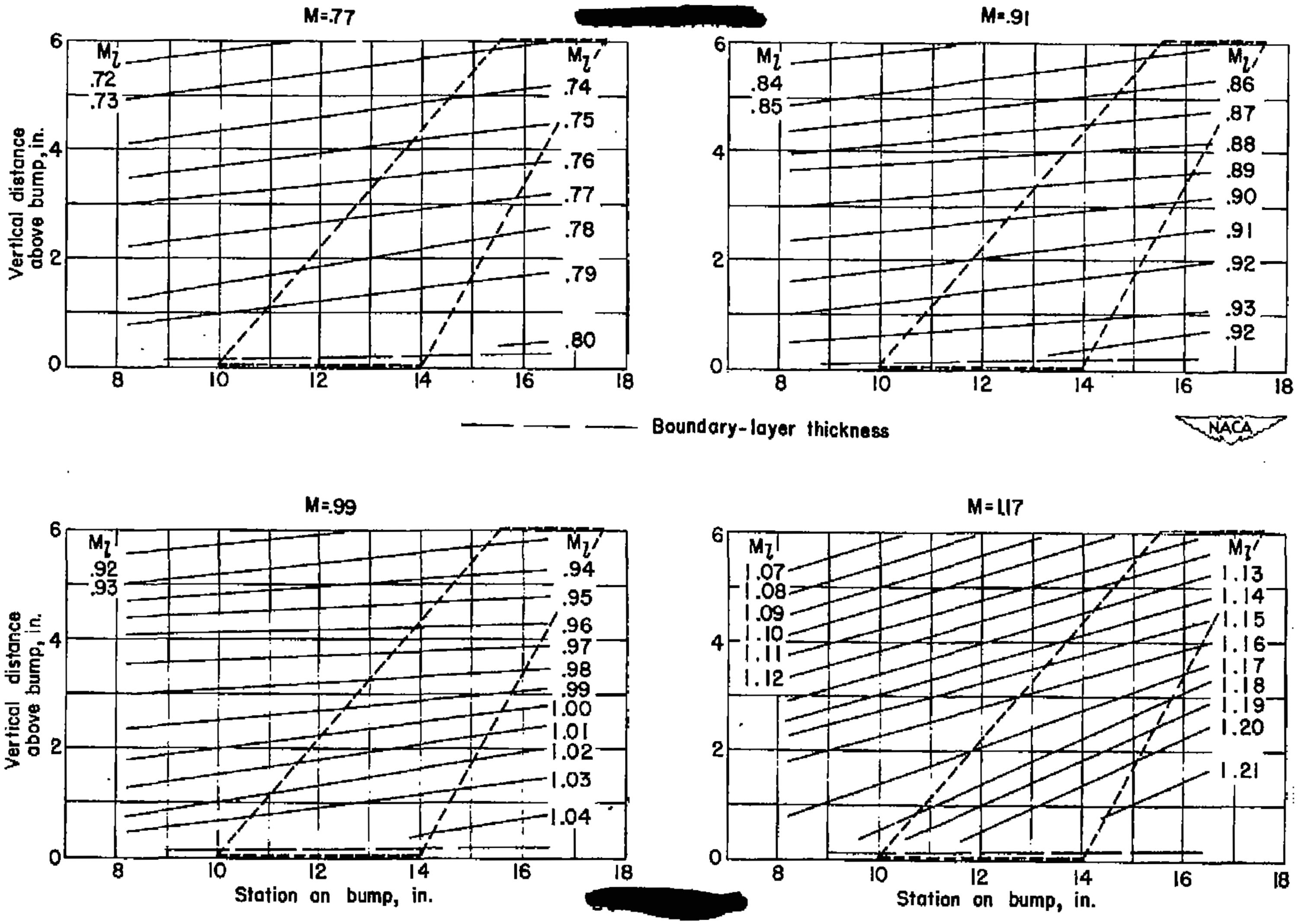

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Figure 4.– Typical Mach number contours over transonic bump in region of model location.

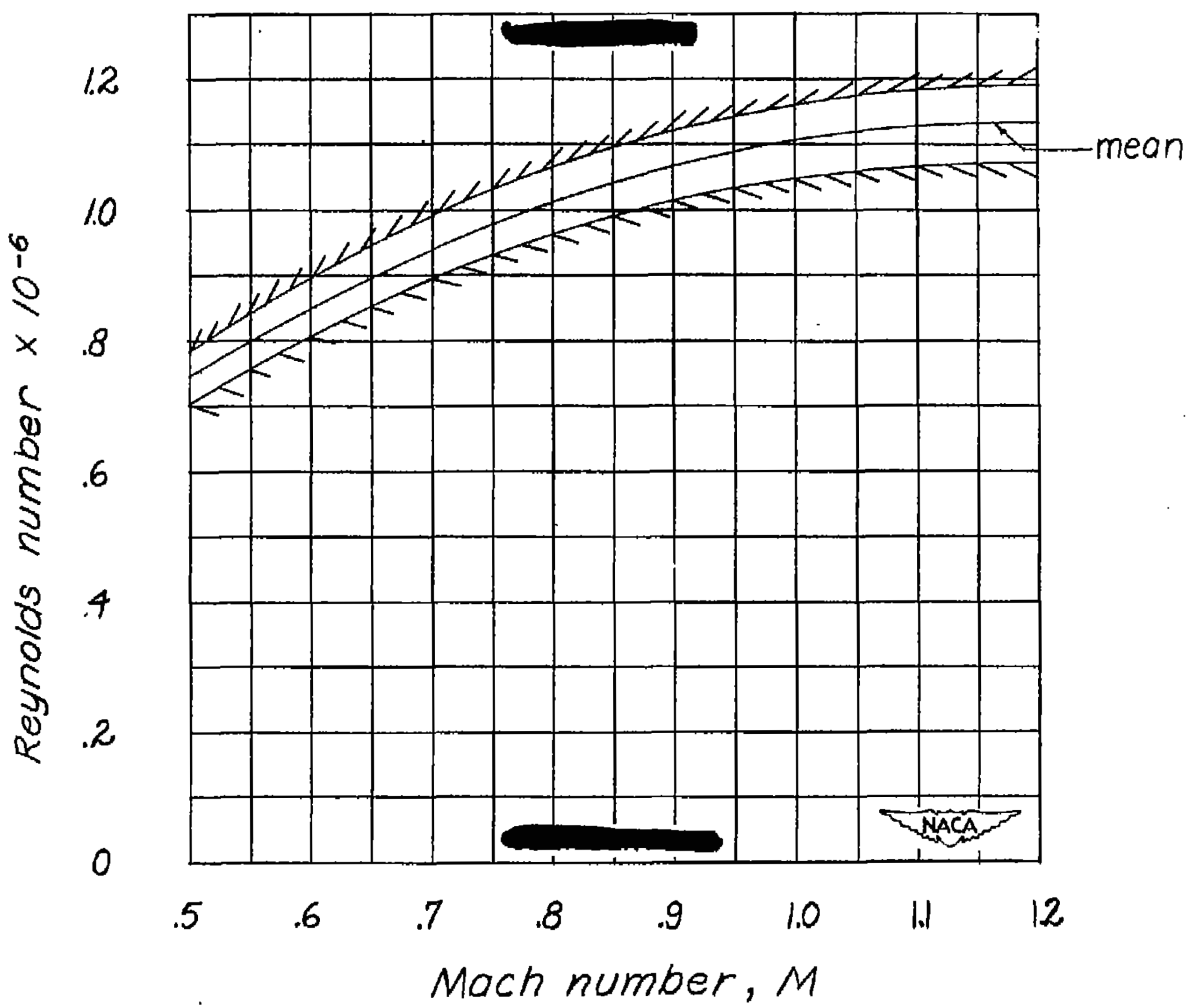

Figure 5.– Variation of Reynolds number with test Mach number through the transonic speed range.

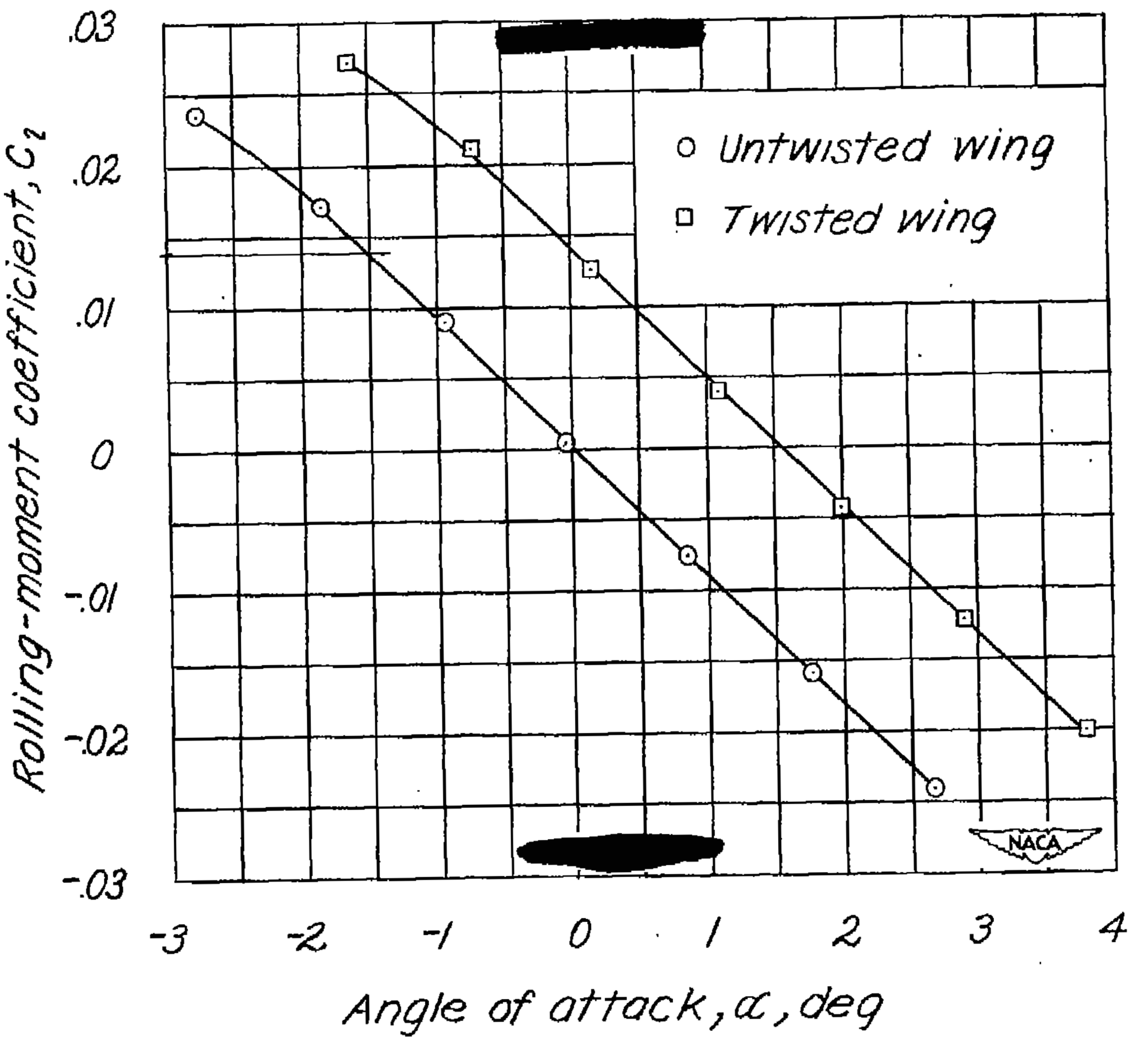

Figure 6.— Variation of rolling-moment coefficient with angle of attack for the plain and twisted wings at a Mach number of 1.90. Circular-arc contour.

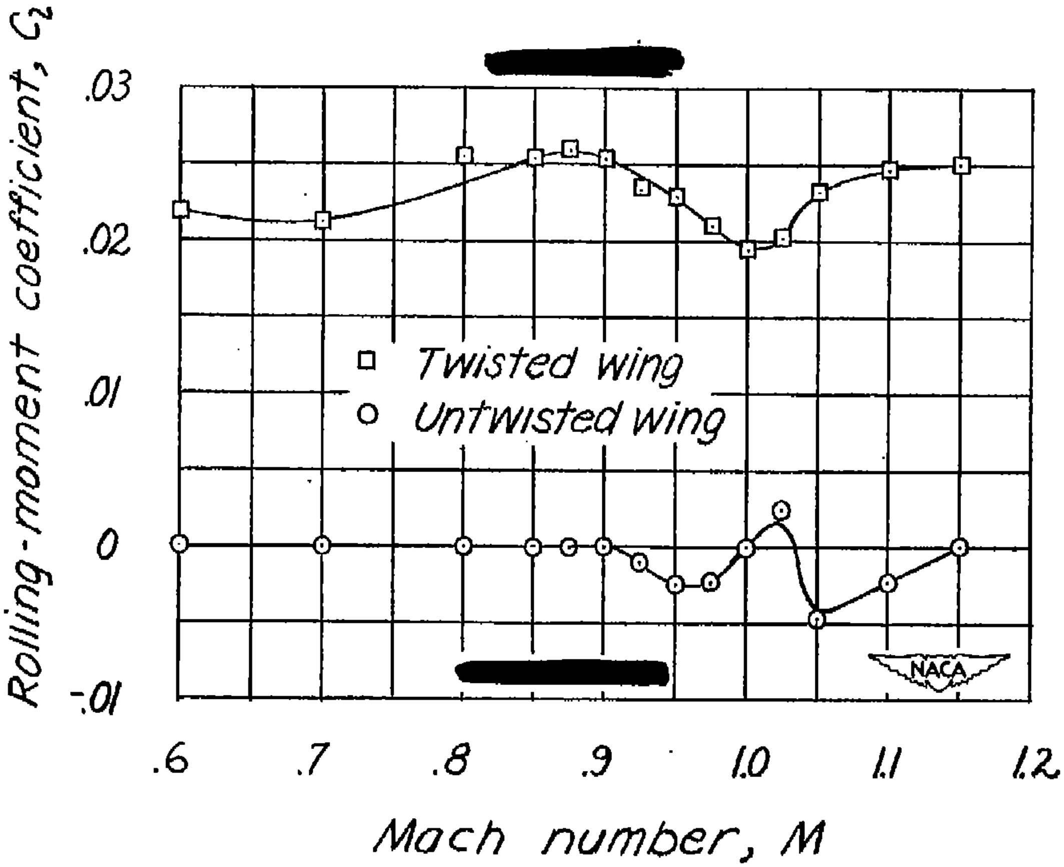

Figure 7.– Typical variation of rolling-moment coefficients through the transonic speed range for the plain and twisted wings. Thickened trailing edge.

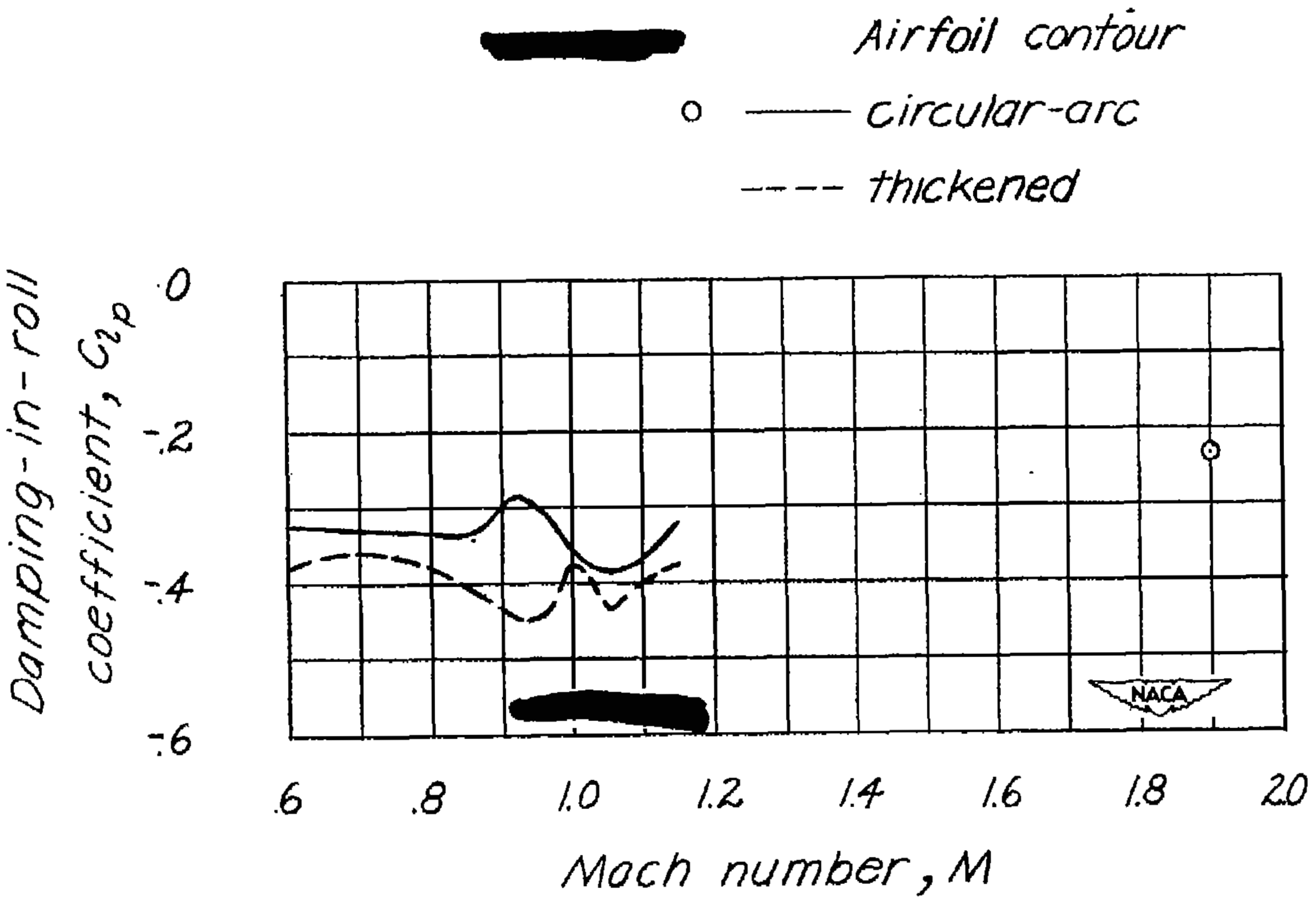

Figure 8.– The effect of airfoil profile on the wing damping-in-roll coefficient.

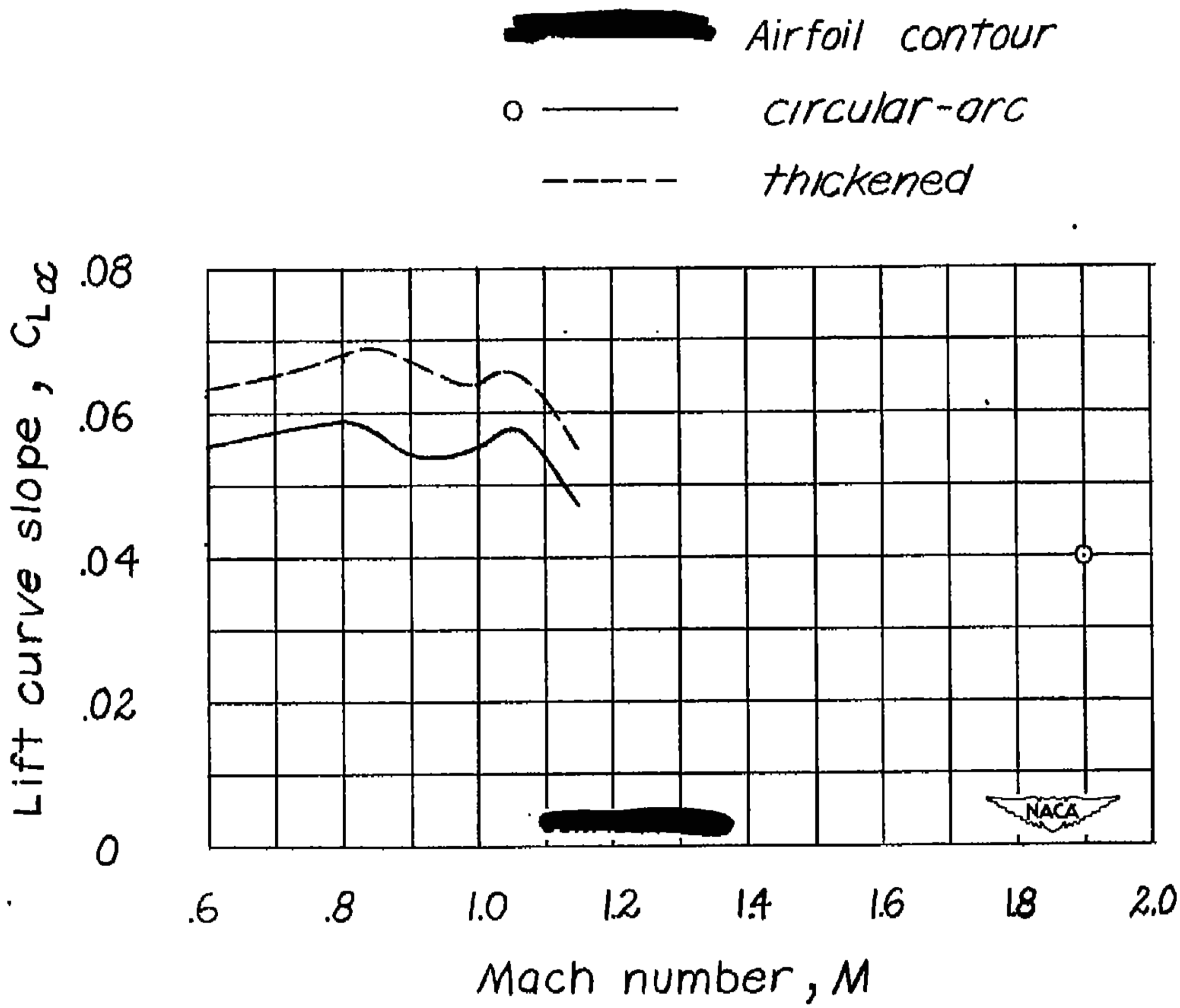

Figure 9.– Effect of airfoil profile on the wing lift-curve slope.

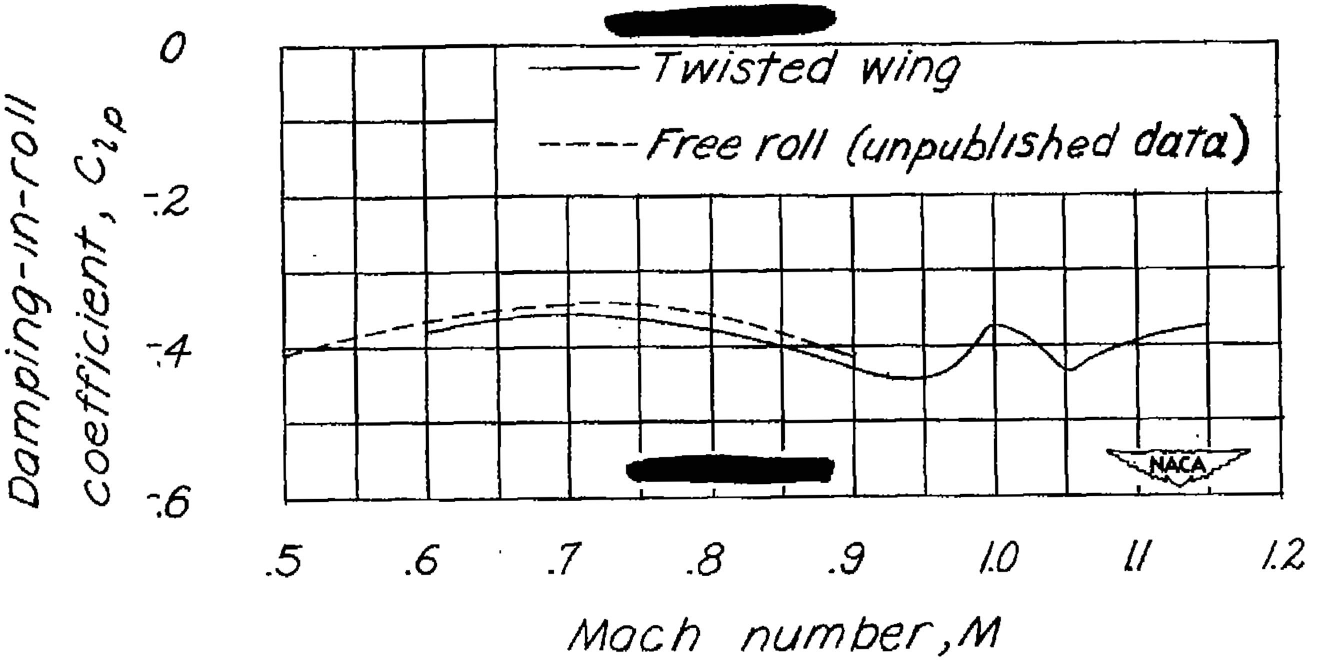

Figure 10.- Comparison of results of two methods of obtaining damping in roll. Thickened trailing edge.

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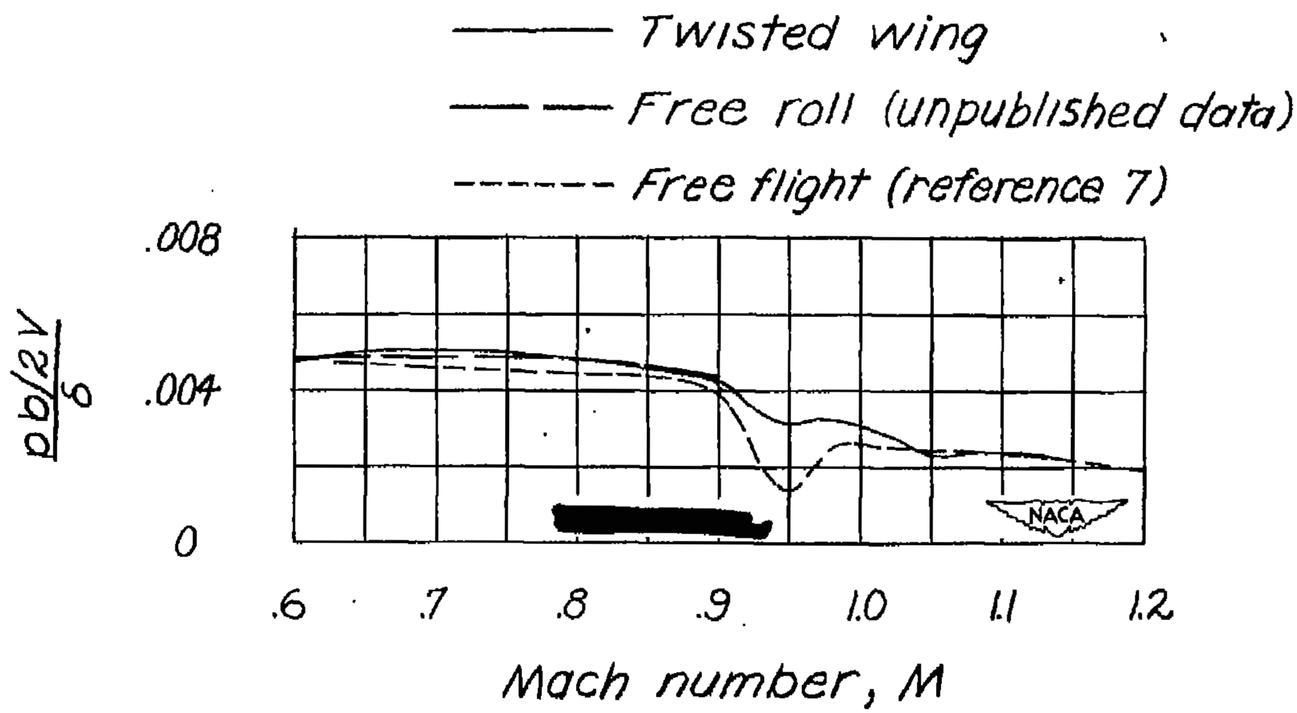

Figure 11.- Comparison of results of three methods of obtaining pb/2V. Thickened trailing edge.

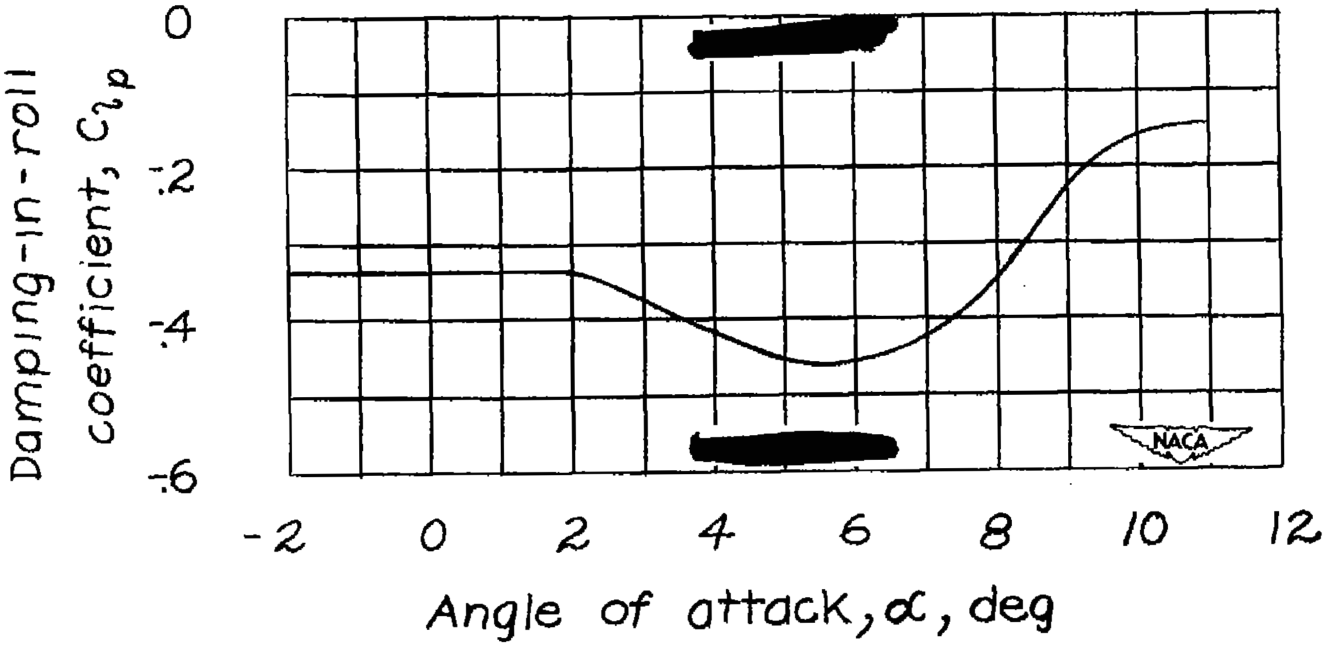

Figure 12.- Variation of the wing-fuselage damping-in-roll coefficient with angle of attack. $M = 0.8$. Thickened trailing edge.

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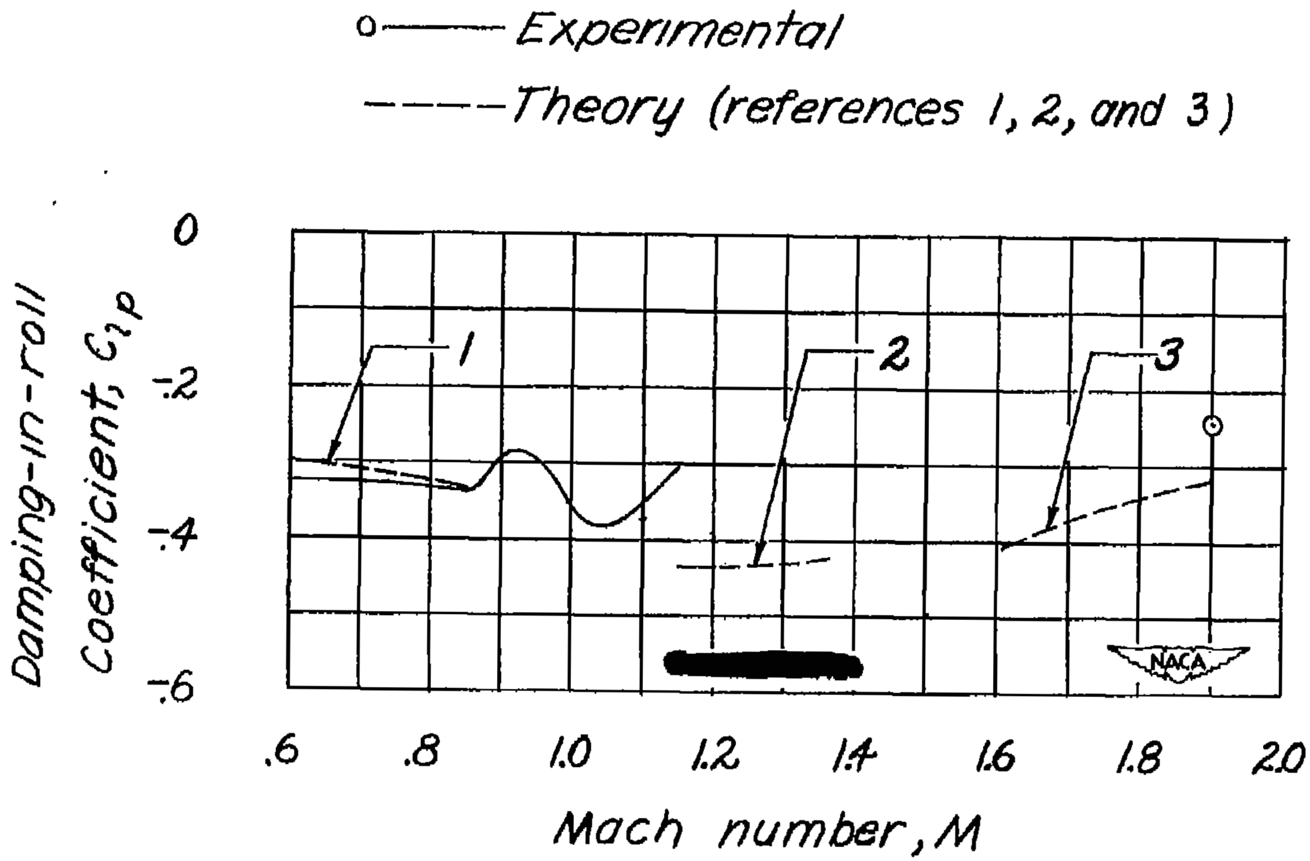

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Figure 13.– Comparison of experimental and theoretical results of damping in roll. Circular-arc airfoil section.